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Note on the performance of propulsive ducts

-by-

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R.A.E. Reference: Eng./E.6/2284/ARH/55

1.0 Introduction

Previous calculations[1] at the R.A.E. on the performance of propulsive ducts had been based on approximate methods as some estimate of their performance was required quickly at the time. Those first calculations were used for the aircraft performance estimates in the note[2] on supersonic flight. The biggest errors involved, as have been pointed out verbally by Rolf and in a note[3] by Griffith, were the neglect of compressibility in the calculations of the pressure losses in the combustion chamber, and, the use of the aircraft total head density instead of the true density at inlet to the combustion chamber, in estimating the mass flow through the propulsive ducts.

This second note gives the results of a more accurate estimation of the performance of propulsive ducts using the same basic losses and efficiencies as in the previous note.

2.0 Method of calculation

2.1 General

The following values of efficiencies and losses were used in the performance estimation.

(a) Intake efficiency 100% for Mach numbers below unity, but with the appropriate shock loss taken into account for Mach numbers greater than unity.

(b) Combustion chamber of constant cross-sectional area, with a total head pressure loss equal, for incompressible flow, to four times the inlet velocity head plus the fundamental loss due to heating. A correction was made for the effect of compressibility.

(c) A combustion fuel loss of 5% corresponding to a chamber with the pressure loss as taken in (b).

(d) A jet adiabatic efficiency of 95%.

The true density at inlet to the combustion chamber was used for estimating the mass flow through the propulsive duct. The effect of compressibility on the losses in the combustion chamber which had been neglected in the previous calculation[1] was also taken into account in the present calculations. To simplify the calculations a constant specific heat was assumed and the small effect of variable specific heat on thrust was neglected, the fuel consumptions though were given an approximate correction for the variation of specific heat. Details of the

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method of calculation are given in the following section and the notation used is shown with the sketch of the propulsive duct in Fig.1.

2.2 Intake

The intake adiabatic efficiency (η_{tot_2}) is based on the total head pressure (P_{tot_2}) after the intake or at inlet to the combustion chamber and is given by

$$\eta_{tot_2} = \frac{\left\{\left(\frac{P_{tot2}}{P_1}\right)^{\frac{\gamma-1}{\gamma}} - 1\right\}}{\left\{T_{tot_1}/T_1 - 1\right\}}$$

where P_1 and T_1 are the pressure and temperature of the ambient air respectively and $T_{tot_1}(= T_{tot_2})$ is the total head temperature equal to T_1 plus the temperature corresponding to the aircraft velocity. For Mach numbers below unity this efficiency is taken to be 100%, but the appropriate plane shock wave losses are taken into account for the aircraft speeds above sonic. A curve of η_{tot_2} against Mach number is given in Fig.2, the values plotted are somewhat different from those quoted in the previous note[1] as the latter were based on static pressures before and after the shock wave. The method of calculation is well known but is given in Appendix I for completeness.

2.3 Combustion chamber

The effect of compressibility on the fundamental loss due to heating and the estimation of choking flows in frictionless combustion chambers can be found by using the principles of momentum as in Ref.3. When aerodynamic losses are introduced by baffles, skin friction, etc., the calculation can only be made on certain simplifying assumptions such as with the aerodynamic drag concentrated at inlet and outlet from the combustion chamber. But even under these conditions the losses and choking flows depend considerably on the manner in which the loss is introduced, such as axial or radial flow with baffles, mixing of streams, skin friction, etc. So that, in the absence of sufficient experimental evidence or more accurate theoretical estimations, a simplified conception of the effects of compressibility has been used in this note.

The incompressible combustion chamber loss equal to 4 inlet velocity heads plus a fundamental loss of $(T_{tot_3}/T_{tot_2} - 1)$ velocity heads is used to determine the size of an equivalent nozzle which will give the same total incompressible loss as the combustion chamber, if it is assumed that all the velocity head at outlet from the nozzle is lost. Keeping the size of the equivalent nozzle fixed, it is then easy to calculate the effect of compressibility on the losses which are equal to the outlet velocity head from the nozzle. The method of calculation and the curve used are given in Appendix II, and Fig.3 respectively. The choking of the flow may occur in the equivalent nozzle or at outlet from the combustion duct depending on the relative values of the aerodynamic and fundamental losses respectively.

In the following table values, calculated by the above method, are given of the inlet Mach number (M_{n2}) to the combustion chamber and of the ratio of the (compressible/incompressible) loss under choking or maximum mass flow conditions for various temperature ratios (T_{tot_3}/T_{tot_2}).

T_{tot_3}/T_{tot_2}	3	4	5	6	
Choking Mn_2	0.23	0.21	0.19	0.18	
Incompressible loss	6	7	8	9	inlet velocity heads
(Comp:/Incomp;)loss	1.47	1.44	1.41	1.39	

The above concept of the effect of compressibility is unsatisfactory for the ideal combustion chamber with no aerodynamic losses, but it appears to give a fairly good approximation for an average low-loss chamber if the aerodynamic losses are comparable with the fundamental losses.

2.4 Jet

The jet conditions are defined by an adiabatic jet efficiency (η_4) of 95% where

$$\eta_4 = \frac{\left\{1 - {}^{T_4}/T_{tot_3}\right\}}{\left\{1 - \left({}^{P_4}/P_{tot_3}\right)^{\frac{\gamma-1}{\gamma}}\right\}}$$

where P_4 and T_4 refer to static conditions at the jet and P_{tot_3} and T_{tot_3} refer to total heads after the combustion chamber. The jet velocity (V_4) is given by the temperature drop

$$(T_{tot_3} - T_4) = V_4^2/2kp.$$

2.5 Overall perfromance

The calculation of the overall performance is straightforward once the losses etc. of the previous sections have been determined. The actual calculations can be carried out in a tabular form as given in Appendix III. It should be noted that the bottom half of Fig.2 gives a curve which is used for the approximate correction of fuel consumption for variable specific heats.

The results of this performance estimation are shown in Figs. 4 to 15 for aircraft speeds from 300 to 1500 m.p.h. and gas temperatures after combustion from 1000 to 2000°C abs.(both ranges quoted being equivalent sea level). At the higher speeds the thrust curves are stopped when there is choking in the combustion chamber. The drags given in the above Figs.4 to 15 are the internal drags of the propulsive ducts when there is no combustion. The intake area is defined as the area at inlet with a velocity of flow equal to the aircraft velocity, other values of course of this velocity of flow will give correspondingly different inlet areas.

The weight of the internal or "engine" parts of propulsive ducts are of the order of 50 to 100 lbs. per square foot of combustion chamber cross-sectional area.

3.0 Comparison with previous calculations

A comparison with the previous calculations [1,2] is shown in Fig.16 for a (combustion chamber inlet velocity/ aircraft velocity) equal to 0.1, this being the value used in the supersonic flight note[2]. The differences involved are not very large so that the conclusions of Ref.2 should not be affected. However, at higher values of combustion chamber inlet velocities the differences become large, especially at the higher aircraft speeds and gas temperatures.

4.0 Conclusions

This note gives the results of a general estimation of the performance of propulsive ducts covering aircraft speeds from 300 to 1500 m.p.h. and gas temperatures after combustion from 1000 to 2000°C.abs.(both equivalent sea level). It is intended to replace some less accurate results given in a previous note where various compressibility effects were neglected.

REFERENCES

No.	Author(s)	Title, etc.
1	Staff of Engine Dept.	A brief discussion on the use and performance of propulsive ducts. R.A.E. Tech.Note No.Eng.159.(June, 1943).
2	R.Smelt, C. M. Fougere and A.R.Howell	Note on supersonic flight. R.A.E. Tech.Note No.Aero 1216 (Flight) (June, 1943).
3	A.A.Griffith	Some considerations concerning propulsive ducts. Rolls Royce (1943).

Attached

Appendices I to III

Print	No.	Eng.	1205	Fig.1	Print	No.	Eng.	1213	Fig.9
"	"	"	1206	" 2	"	"	"	1214	" 10
"	"	"	1207	" 3	"	"	"	1215	" 11
"	"	"	1208	" 4	"	"	"	1216	" 12
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APPENDIX I

Intake efficiency

For intake Mach numbers lower than 1.0 the intake total head efficiency was taken to be 100%.

For intake Mach numbers higher than 1.0 the intake total head efficiency was calculated and is plotted in Fig.2 against inlet Mach number.

To do this it was first necessary to obtain M_2, the Mach number after the intake, in terms of M_1, the Mach number of the air before the shock wave. Having done this, the total head efficiency was obtained in terms of M_1 and M_2 and could then be plotted against M_1.

It should be noted that the suffix "n" on M_n for Mach number has not been used in this Appendix and that the suffix 2 refers to conditions immediately after the shock and not to those at inlet to the combustion chamber.

Considering continuity, the mass flow before and after the shock wave must be the same, and the area is also the same

$$\therefore \quad A\,\rho_1 V_1 = A\,\rho_2 V_2$$

$$\text{or} \quad \rho_1 V_1 = \rho_2 V_2$$

$$\therefore \quad \frac{V_1}{V_2} = \frac{\rho_2}{\rho_1} = \frac{P_2}{P_1} \times \frac{T_1}{T_2}$$

$$\text{giving} \quad \frac{P_2}{P_1} = \frac{V_1 T_2}{V_2 T_1}$$

$$\text{Then since} \quad M = \frac{V}{\sqrt{\gamma KT}}$$

$$\frac{P_2}{P_1} = \frac{M_1}{M_2}\sqrt{\frac{T_2}{T_1}} \qquad \dots\dots\dots\dots\dots\dots \qquad (1)$$

Assuming frictionless flow, the force on a plane before the shock wave will be the same as the force on a plane after the shock wave.

$$\therefore \quad P_1 A + \omega V_1 = P_2 A + \omega V_2$$

$$\text{or} \quad P_1 + \rho_1 V_1^2 = P_2 + \rho_2 V_2^2$$

$$\text{and} \quad P_1 + \frac{P_1}{KT_1} \times V_1^2 = P_2 + \frac{P_2}{KT_2} \times V_2^2$$

$$\therefore \quad P_1\left(1 + \frac{V_1^2}{KT_1}\right) = P_2\left(1 + \frac{V_2^2}{KT_2}\right)$$

$$\therefore \quad P_1\left(1 + \gamma M_1^2\right) = P_2\left(1 + \gamma M_2\right)^2$$

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$$\text{or} \quad \frac{P_2}{P_1} = \frac{1 + \gamma M_1^2}{1 + \gamma M_2^2} \quad \ldots\ldots\ldots\ldots\ldots\ldots \quad (2)$$

from conservation of energy

$$K_p T_{tot} = K_p T_1 + \tfrac{1}{2} V_1^2 = K_p T_2 + \tfrac{1}{2} V_2^2$$

$$\frac{\gamma}{\gamma-1} K\, T_{tot} = \frac{\gamma}{\gamma-1} KT_1 + \tfrac{1}{2} M_1^2\, \gamma KT_1 = \frac{\gamma}{\gamma-1} KT_2 + \tfrac{1}{2} M_2^2\, \gamma KT_2$$

$$\therefore T_{tot} = T_1\left(1 + \frac{\gamma-1}{2} M_1^2\right) = T_2\left(1 + \frac{\gamma-1}{2} M_2^2\right) \quad \ldots\ldots \quad (3)$$

$$\text{or} \quad \frac{T_2}{T_1} = \frac{1 + \frac{\gamma-1}{2} M_1^2}{1 + \frac{\gamma-1}{2} M_2^2} \quad \ldots\ldots\ldots\ldots\ldots\ldots \quad (4)$$

From equations (1), (2) and (4)

$$M_2 = \sqrt{\frac{1 + \frac{\gamma-1}{2} M_1^2}{\gamma^2 M_1^2 - (\gamma-1)\gamma M_1^2 - \frac{\gamma-1}{2}}}$$

It now remains to obtain the total head intake efficiency,

$$\eta_{tot_2} = \frac{\left\{\left(\frac{P_{tot_2}}{P_1}\right)^{\frac{\gamma-1}{\gamma}} - 1\right\}}{\frac{T_{tot_1}}{T_1} - 1} \quad \text{in terms of } M_1 \text{ and } M_2.$$

$$\left(\frac{P_{tot_2}}{P_2}\right)^{\frac{\gamma-1}{\gamma}} = \frac{T_{tot_2}}{T_2}$$

$$\left(\frac{P_{tot_2}}{P_1}\right)^{\frac{\gamma-1}{\gamma}} = \frac{T_{tot_2}}{T_2} \times \left(\frac{P_2}{P_1}\right)^{\frac{\gamma-1}{\gamma}}$$

$$= \left(1 + \frac{\gamma-1}{2} M_2^2\right) \times \left(\frac{1 + \gamma M_1^2}{1 + \gamma M_2^2}\right)^{\frac{\gamma-1}{\gamma}} \quad \text{from equations (2) and (3).}$$

$$\frac{T_{tot_1}}{T_1} - 1 = \frac{\gamma-1}{2} M_1^2 \quad \text{from equation (3)}$$

Giving the total head intake efficiency η_{tot_2} in terms of M_1 and M_2 as:-

$$\eta_{tot_2} = \frac{\left\{\left(1 + \frac{\gamma-1}{2} M_2^2\right) \times \left(\frac{1 + \gamma M_1^2}{1 + \gamma M_2^2}\right)^{\frac{\gamma-1}{\gamma}} - 1\right\}}{\frac{\gamma-1}{2} \cdot M_1^2}$$

APPENDIX II

Combustion chamber losses

It was necessary to find a method of calculating losses of total head pressure in the combustion chamber which would apply equally to compressible and incompressible flow.

A method of doing this appeared to be to consider the flow as being in an equivalent nozzle of discharge area a (x combustion chamber area), giving the required losses for incompressible flow and which would also apply to compressible flow.

Incompressible flow:-

$$\text{Total head loss} = \left\{\left(\frac{T_{tot_3}}{T_{tot_2}} - 1\right) + 4\right\} \times (\text{inlet velocity head})$$

where $\left(\frac{T_{tot_3}}{T_{tot_2}} - 1\right) \times$ (inlet velocity head) is fundamental loss due to heating

and 4 x (inlet velocity head) is aerodynamic pressure loss

$$\text{or } \frac{\text{Total head loss}}{\text{inlet velocity head}} = \frac{P_{tot_2} - P_{tot_3}}{\frac{1}{2}\rho V_2^2} = \frac{T_{tot_3}}{T_{tot_2}} + 3 \quad \ldots\ldots\ldots\ldots (1)$$

For an equivalent nozzle of discharge area a to give the same losses as a combustion chamber of area A:-

$$P_{tot_2} - P_{tot_3} = P_{tot_2} - P_n \quad \text{where } P_n \text{ is static pressure in nozzle.}$$

$$= \tfrac{1}{2}\rho V_n^2 \quad \text{where } V_n \text{ is velocity in nozzle}$$

$$= \tfrac{1}{2}\rho V_2^2 \times \frac{1}{a^2}$$

$$\therefore \frac{\text{Total head loss in nozzle}}{\frac{1}{2}\rho V_2^2} = \frac{1}{a^2}$$

For the loss in total head in the nozzle to be equal to the loss in the combustion chamber given by equation (1)

$$a = \sqrt{\frac{1}{\frac{T_{tot_3}}{T_{tot_2}} + 3}}$$

With 'standard entry conditions' (14.7 lb./sq.in. and 288° C.abs) and values of V_n from 0 to 1000 ft./second, ΔT, the temperature drop in the equivalent nozzle, is calculated from the formula:-

$$\Delta T = \frac{V^2}{2K_p} = 0.461 \left(\frac{V}{100}\right)^2 \quad \text{where } \Delta T \text{ is in °C and V is in ft./sec.}$$

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The pressure drop ΔP is calculated for each value of V from:-

$$1 - \frac{\Delta P}{14.7} = \left\{1 - \frac{\Delta T}{288}\right\}^{\frac{\gamma}{\gamma-1}}$$

$$\text{giving } \Delta P = 14.7\left\{1 - \left(1 - \frac{\Delta T}{288}\right)^{3.51}\right\} \text{ lb./sq.in.}$$

The relative density σ for each value is given by:-

$$\sigma = \frac{14.7 - \Delta P}{14.7} \times \frac{288}{288 - \Delta T}$$

$$\text{since } \frac{V}{100} = \frac{18.8\,\omega}{\sigma A}$$

$$\frac{\omega}{A} = \frac{\frac{V}{100} \times \sigma}{18.8}$$

where ω is mass flow in lb/sec
A is area in sq.in.
σ is relative density.

In order that calculations may be done from inlet conditions of T_{tot_2} and P_{tot_2}, the values for V_n are plotted in fig.3, as $V_n \times \sqrt{\frac{288}{T_{tot_2}}}$ and for ΔP as $\Delta P \times \frac{14.7}{P_{tot_2}}$.

$\frac{\omega}{A}$ is plotted as $\frac{\omega}{A} \times \frac{14.7}{P_{tot_2}} \times \sqrt{\frac{T_{tot_2}}{288}}$

Choosing a value of V_n through the equivalent nozzle $V_n\sqrt{\frac{288}{T_{tot_2}}}$ is calculated and $\Delta P \times \frac{14.7}{P_{tot_2}}$ can be read from fig.3.

Then $(1 - \frac{\Delta P}{P_{tot_2}}) \times \frac{P_{tot_2}}{14.7}$ gives the pressure ratio after the combustion chamber and, from the corresponding value of $\frac{\omega}{aA} \times \frac{14.7}{P_{tot_2}} \times \sqrt{\frac{T_{tot_2}}{288}}$, the mass flow ω through the equivalent nozzle and through the combustion chamber is calculated.

Pd

APPENDIX III

Overall performance calculation

Separate performance calculations have to be made for each aircraft speed required.

The temperature rise ΔT due to the appropriate aircraft velocity is obtained from $\Delta T = \frac{V_1^2}{2K_p} = .461 \left(\frac{V_1}{100}\right)^2$ where V_1 is the velocity in f.p.s.

The total head pressure ratio is obtained from

$$R_2 = \frac{P_{tot_2}}{14.7} = \left(1 + \eta_{tot_2}\frac{\Delta T}{288}\right)^{3.51}$$

where η_{tot_2} is the intake efficiency. (Appendix I).

The relative density δ_2 at intake is $\frac{P_{tot_2}}{14.7} \times \frac{288}{T_{tot_2}}$

The calculations were performed for temperatures T_{tot_3} after combustion of T_{tot_2}, 1000, 1500 and 2000°C.abs.

For each aircraft speed four values of $V_n \frac{288}{T_{tot_2}}$ were chosen and calculations at each T_{tot_3} were based on these values.

$$\text{Thrust} = \frac{V_4 - V_1}{g} \times \omega$$

where V_4 = jet velocity in f.p.s.
V_1 = aircraft velocity in f.p.s.
ω = mass flow in lb./sec.

Using fuel of a calorific value of 10,300 C.H.U. and allowing for 5% not being burnt

$$\text{consumption} = 0.0843 \times \frac{100}{95} \times (T_{tot_3} - T_{tot_2}) \times \omega \times \left(\frac{\text{Mean spec. heat}}{0.241}\right)$$

where the final term $\left(\frac{\text{mean specific heat}}{0.241}\right)$ is read, for the appropriate temperature after combustion, from fig.2.

$$\text{Area of jet} = \frac{18.8 \times \omega \times 100}{\delta_4 \times V_4 \times 144} = \frac{13.05 \times \omega}{\delta_4 V_4}$$

where δ_4 is relative density of jet.

$$\text{Overall efficiency} = \frac{\frac{\text{Thrust}}{\text{consumption}} \times \frac{\text{aircraft speed in mph.}}{100} \times \frac{88}{60} \times 3600 \times 10{,}000}{10{,}300 \times 1{,}400}$$

$$= \frac{3.66 \times \frac{\text{aircraft speed in m.p.h.}}{100}}{\text{specific fuel consumption}}$$

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When $\frac{\omega}{A} \times \frac{14.7}{[illegible]} \times \sqrt{\frac{T_{tot_3}}{288}}$ is greater than .345 [illegible]ach number at outlet [illegible] the combustion [illegible]amber is 1.0 and the flow i[illegible]en to be choking, so the curves are stopped if the mass flow reaches this val[illegible]

The actual performance calculations at each aircraft speed were done in the following tabular form for sea level conditions.

Intake η_{tot_2} = (fig.2). Velocity temp. rise $\Delta T = .461\left(\frac{V_1}{100}\right)^2 =$

$$\frac{P_{tot_2}}{14.7} = \left(1 + \eta_{tot_2}\frac{\Delta T}{288}\right)^{3.51} = R_2 = . \; \delta_2 =$$

1	T_{tot_3} °C.abs.	T_{tot_2} (no combustion)	1000	1500	2000
2	a	$1/\sqrt{(T_{tot_3}/T_{tot_2})+3}$			
3	$V_n\sqrt{\frac{288}{T_{tot_2}}}$	Values chosen			
4	V_n	$(3) \times \sqrt{\frac{T_{tot_2}}{288}}$			
5	$\frac{\omega}{aA} \cdot \frac{1}{R} \cdot \sqrt{\frac{T_{tot_2}}{288}}$	From fig.3			
6	$\frac{\omega}{A} \cdot \frac{1}{R} \cdot \frac{T_{tot_2}}{288}$	(5) x (2)			
7	$V_2\sqrt{\frac{288}{T_{tot_2}}}$	Read from fig.3 to correspond with (6)			
8	V_2	$(7) \times \sqrt{\frac{T_{tot_2}}{288}}$			
9	V_2/aircraft speed in f.p.s.	(8)/aircraft speed in f.p.s.			
10	ω lb./sec.	$(6) \times \sqrt{\frac{288}{T_{tot_2}}} \times R \times 144$			
11	$\Delta P \times \frac{14.7}{P_{tot_2}}$	From fig.3 to correspond with (3)			
12	$\Delta P/P_{tot_2}$	(11)/14.7			
13	$1 - \Delta P/P_{tot_2}$	1 - (12)			
14	Jet pressure ratio. R_3	$(13) \times R_2$			
15	Adiabatic temp.ratio jet	$(14)^{.285}$			
16	Adiabatic temp.drop jet °C	$\frac{(15) - 1}{(15)} \times T_{tot_3}$			
17	Actual temp.drop jet °C	.95(16) (η_4 = 95%)			
18	V_4 ft./sec.	$100\sqrt{\frac{(17)}{.461}}$			

.... contd.

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1	T_{tot_3} °C.abs.	T_{tot_2}(no combustion)	1000	1500	2000
19	V_1 ft./sec.	Aircraft velocity in ft./sec.			
20	$V_4 - V_1$ ft./sec.	(18) - (19)			
21	Thrust - lb./sq.ft.c.ch. area	$\frac{(20)}{32.2} \times (10)$			
22	Consumption lb./hr.	$0.0887 \times (T_{tot_3} - T_{tot_2}) \times (10) \times \frac{\text{mean specific heat}}{0.241}$ (from fig.2)			
23	Specific consumption lb./hr./lb thrust	(22)/(21)			
24	Temperature jet °C abs.	(1) - (17)			
25	Relative density jet	288/(24)			
26	Area jet sq.ft.	$\frac{13.05 \times (10)}{(25) \times (18)}$			
27	M_{n_4} jet	$\frac{(18)}{1120} \times \sqrt{(25)}$			
28	Overall efficiency	$\frac{3.66 \times \text{aircraft speed in m.p.h.}/100}{(23)}$			
29	$\frac{\omega}{A} \cdot \frac{14.7}{P_{tot_3}} \cdot \sqrt{\frac{T_{tot_3}}{288}}$	$\frac{(10)}{144} \cdot \frac{1}{(14)} \cdot \sqrt{\frac{(1)}{288}}$			
30	Area inlet/area c.ch.	$\frac{13.05 \times (10)}{1 \times \text{aircraft velocity in ft./sec.}}$			

FIG. 1

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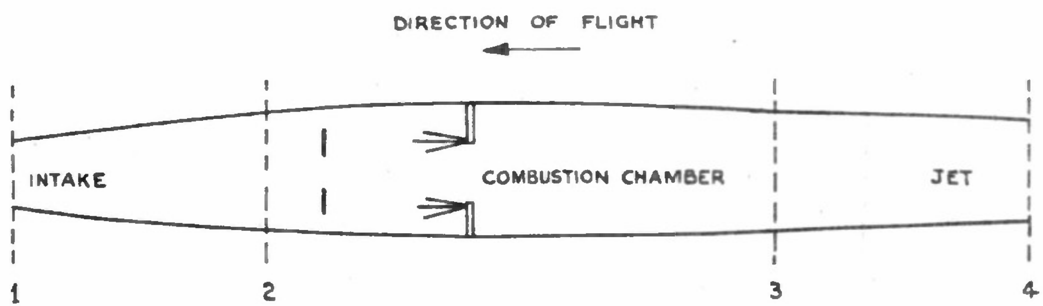
PROPULSIVE DUCT NOTATION



P = STATIC PRESSURE

P_{tot} = TOTAL HEAD PRESSURE

η = EFFICIENCY

T = STATIC TEMPERATURE

T_{tot} = TOTAL HEAD TEMPERATURE

W = MASS FLOW

A = CROSS-SECTIONAL AREA

V = VELOCITY

M_n = MACH NUMBER

SUFFIX 1 DENOTES FLIGHT INLET CONDITIONS

SUFFIX 2 DENOTES INLET TO COMBUSTION CHAMBER

SUFFIX 3 DENOTES OUTLET FROM COMBUSTION CHAMBER

SUFFIX 4 DENOTES OUTLET FROM JET

PROPULSIVE DUCTS

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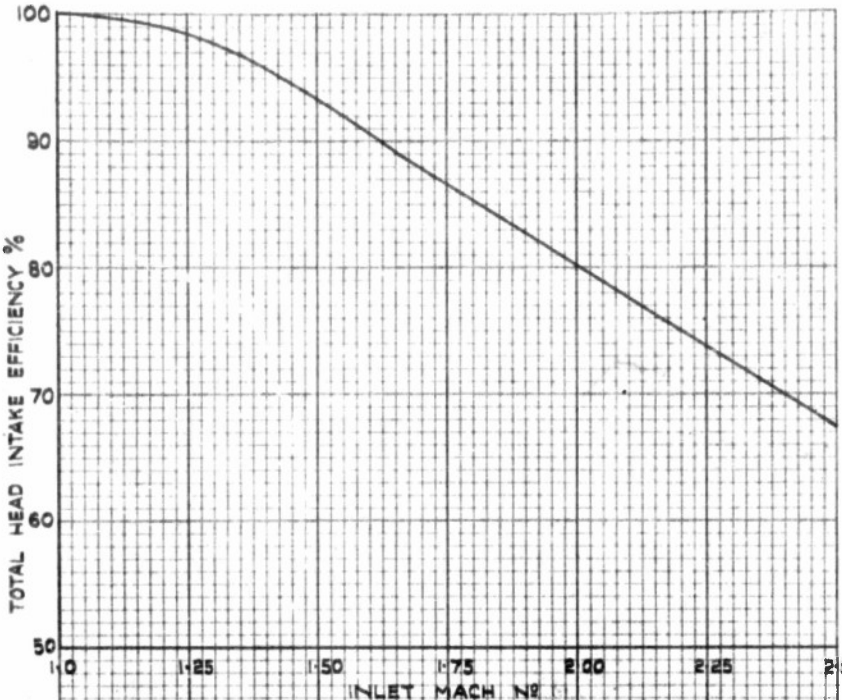
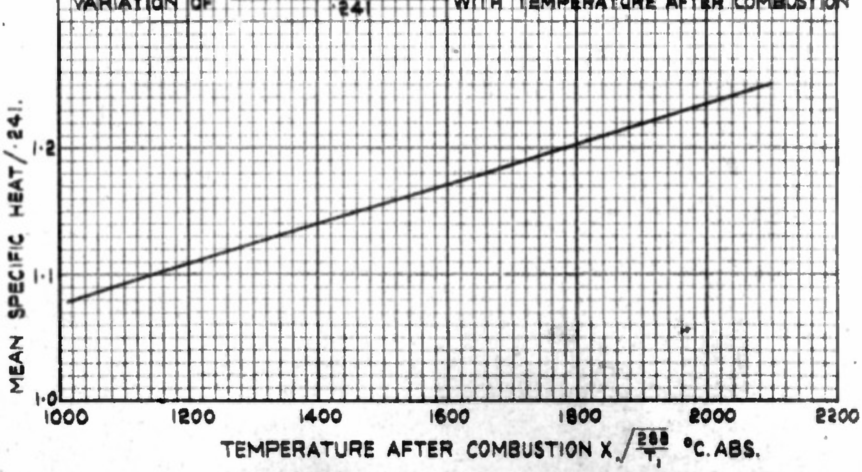
FIG. 2

VARIATION OF TOTAL HEAD INTAKE EFFICIENCY WITH INLET MACH. Nº.



VARIATION OF $\frac{\text{MEAN SPECIFIC HEAT}}{\cdot 241}$ WITH TEMPERATURE AFTER COMBUSTION



PROPULSIVE DUCTS.

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Fig. 3.

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CURVES FOR ESTIMATING COMBUSTION CHAMBER LOSSES

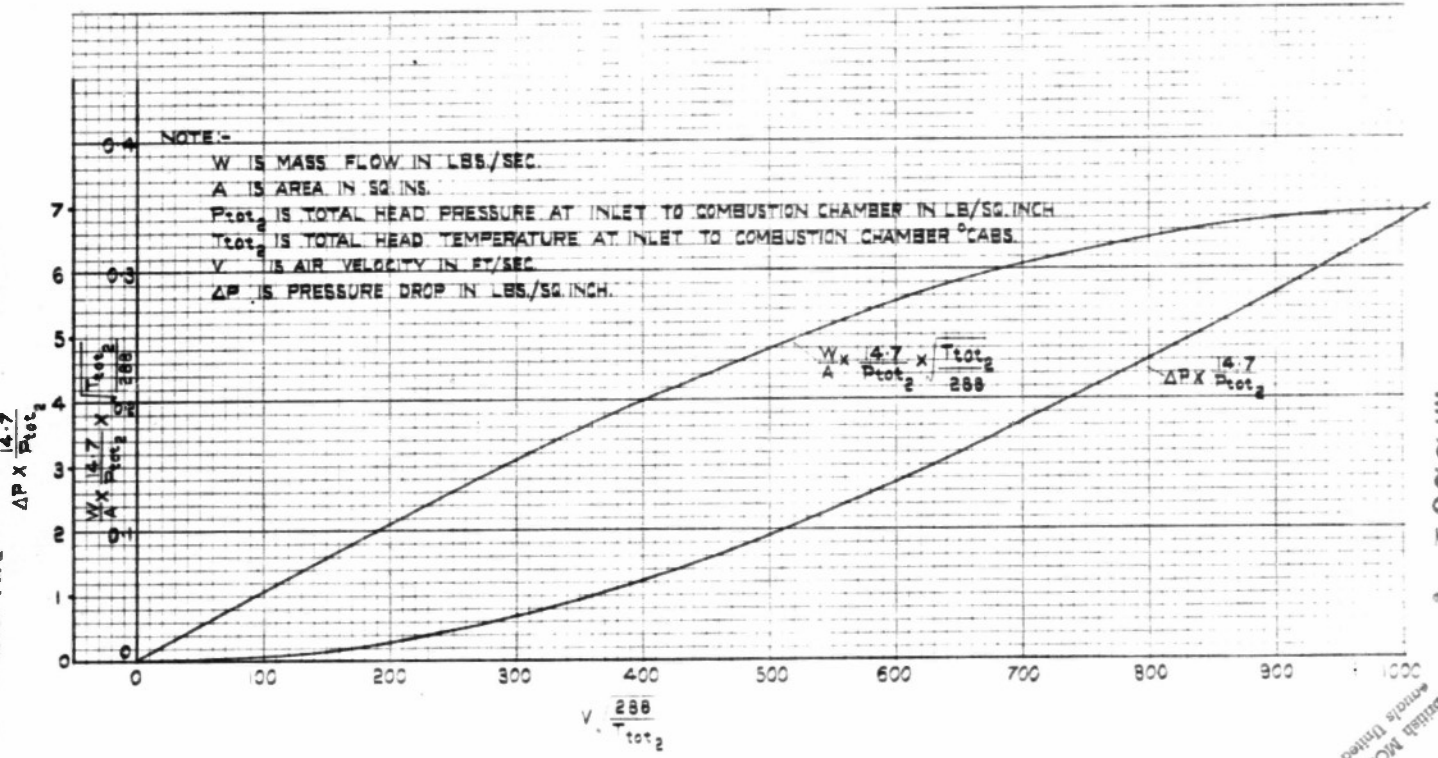

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FIG. 4.

AIRCRAFT VELOCITY = 300 $\sqrt{\frac{T_1}{288}}$ M.P.H.

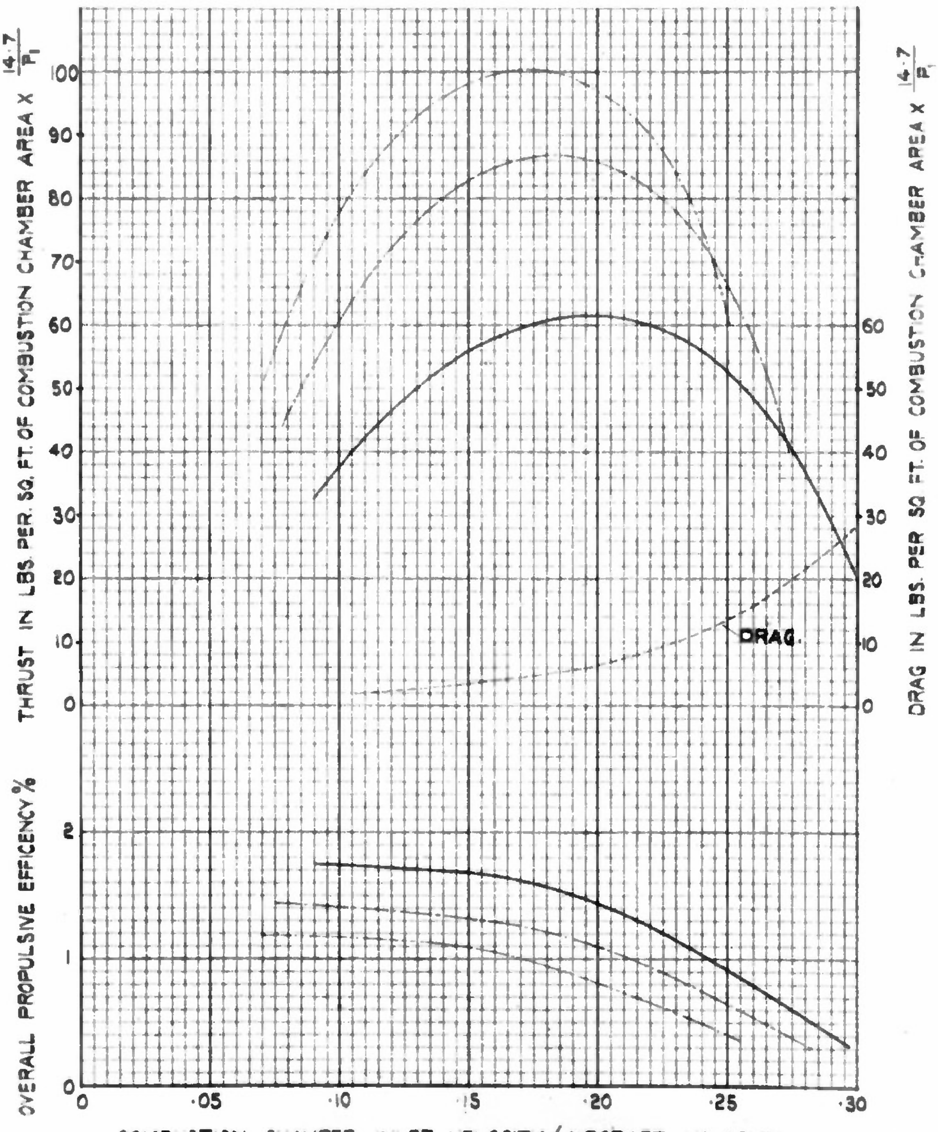
TEMPERATURE AFTER COMBUSTION $1000 \times \frac{T_1}{288}$ °C. ABS.

" " " $1500 \times \frac{T_1}{288}$ °C. ABS.

" " " $2000 \times \frac{T_1}{288}$ °C. ABS.

DRAG WITH NO COMBUSTION

NOTE:- P_1 AND T_1 ARE THE PRESSURE IN LBS./SQ. INCH AND TEMPERATURE IN °C. ABS OF THE AMBIENT AIR.



COMBUSTION CHAMBER INLET VELOCITY / AIRCRAFT VELOCITY.

PROPULSIVE DUCTS.

15d

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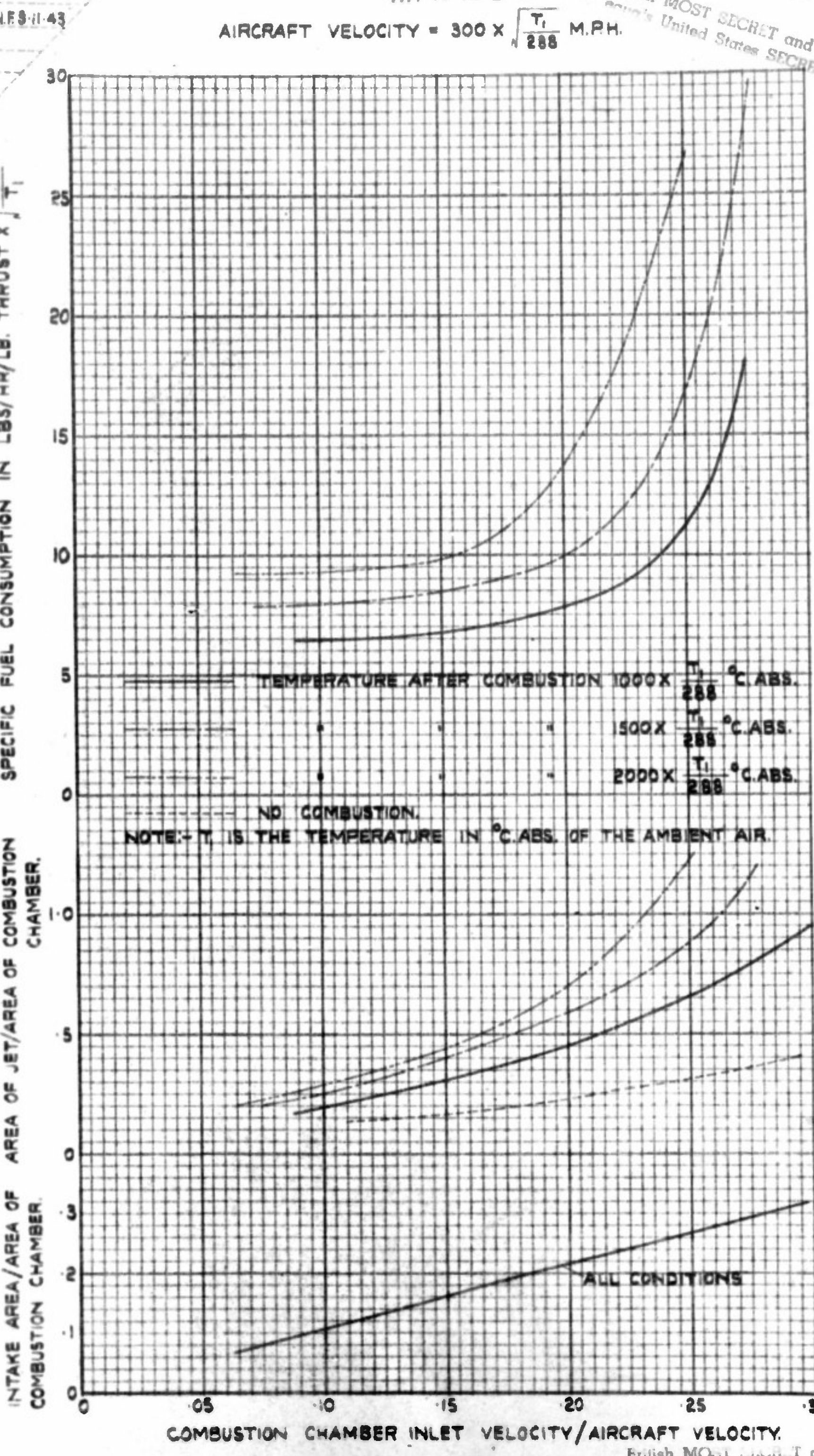
WA-2261 7

FIG. 5.



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WA-22617 Fig. 6

AIRCRAFT VELOCITY = $450 \times \sqrt{\frac{T_1}{288}}$ M.P.H.

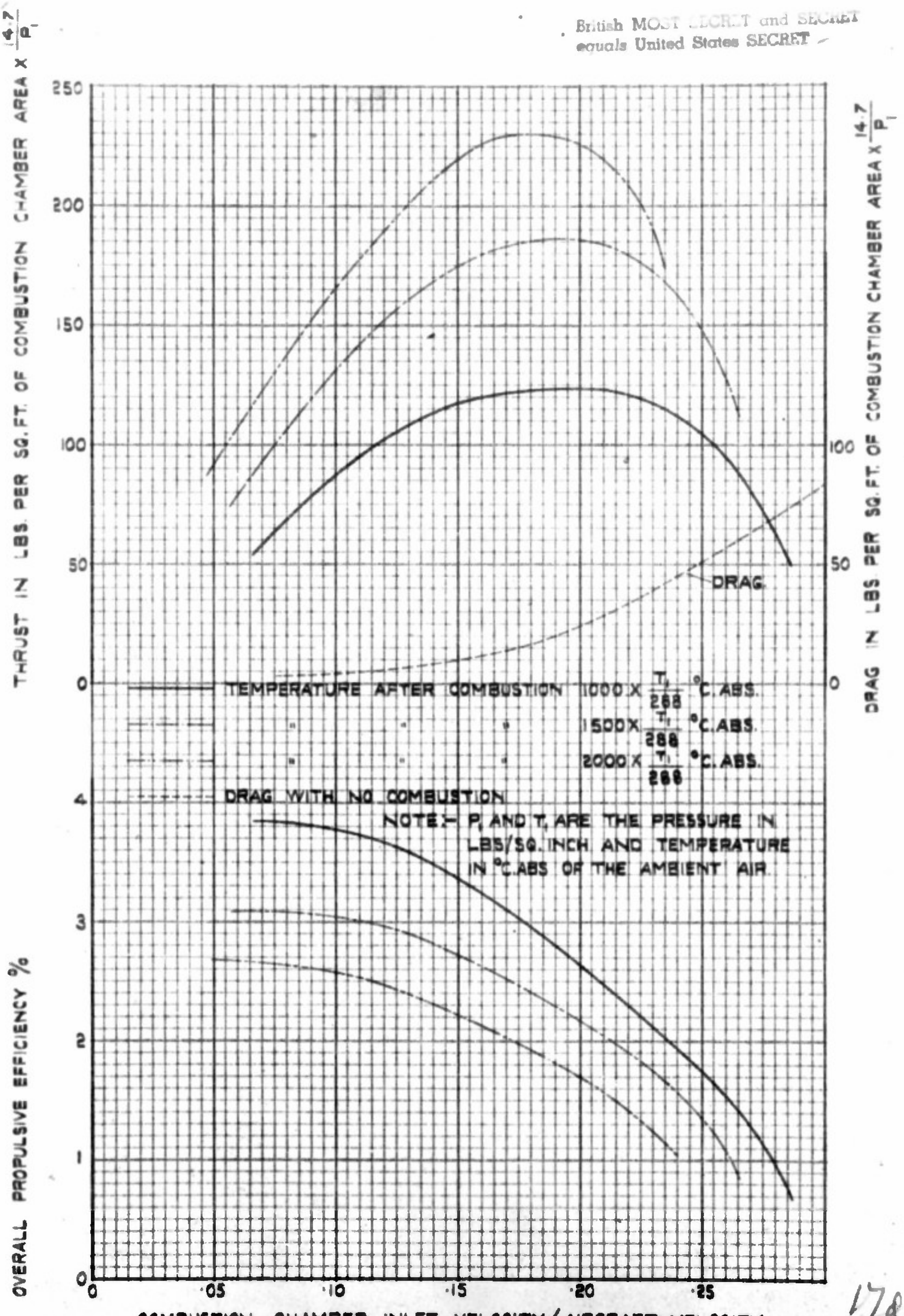

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FIG. 7.

AIRCRAFT VELOCITY = $450 \times \sqrt{\frac{T_1}{288}}$ M.P.H.

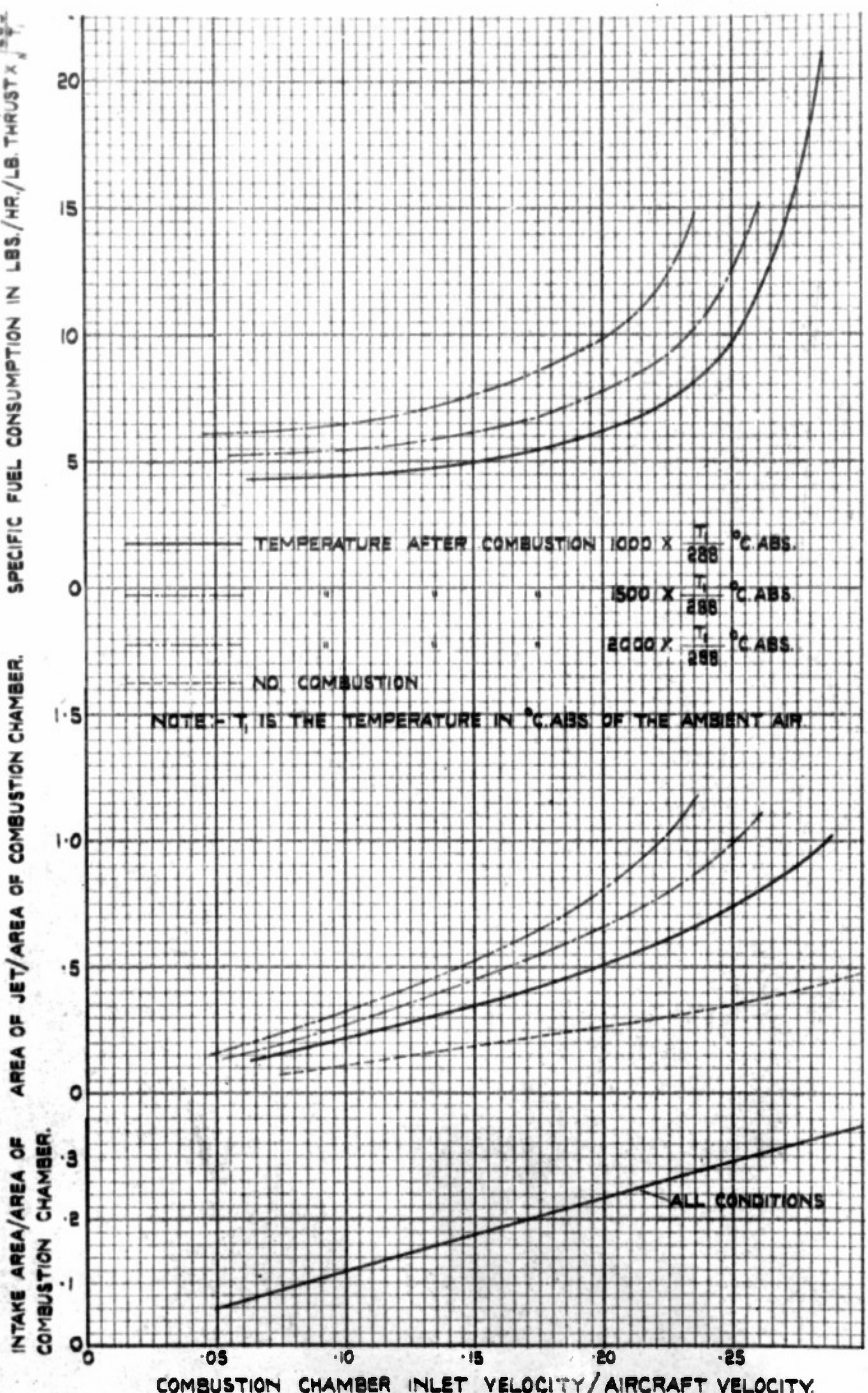

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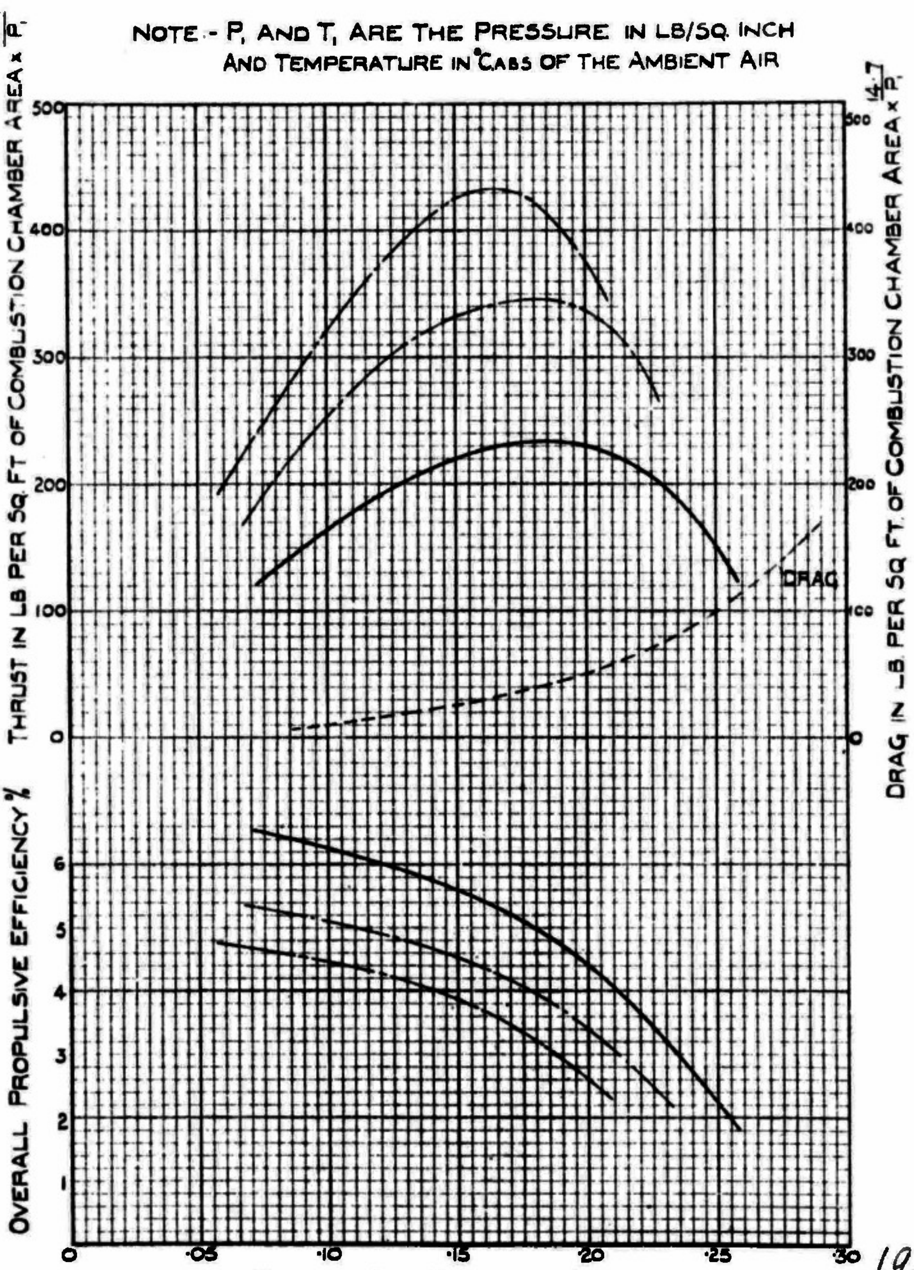
FIG. 8

AIRCRAFT VELOCITY = $600 \times \sqrt{\frac{T_1}{288}}$ M.P.H.

TEMPERATURE AFTER COMBUSTION $1000 \times \frac{T_1}{288}$ °C ABS
" " " $1500 \times \frac{T_1}{288}$ °C ABS
" " " $2000 \times \frac{T_1}{288}$ °C ABS
DRAG WITH NO COMBUSTION

NOTE - P_1 AND T_1 ARE THE PRESSURE IN LB/SQ INCH AND TEMPERATURE IN °C ABS OF THE AMBIENT AIR



COMBUSTION CHAMBER INLET VELOCITY/AIRCRAFT VELOCITY

19d

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FIG. 9

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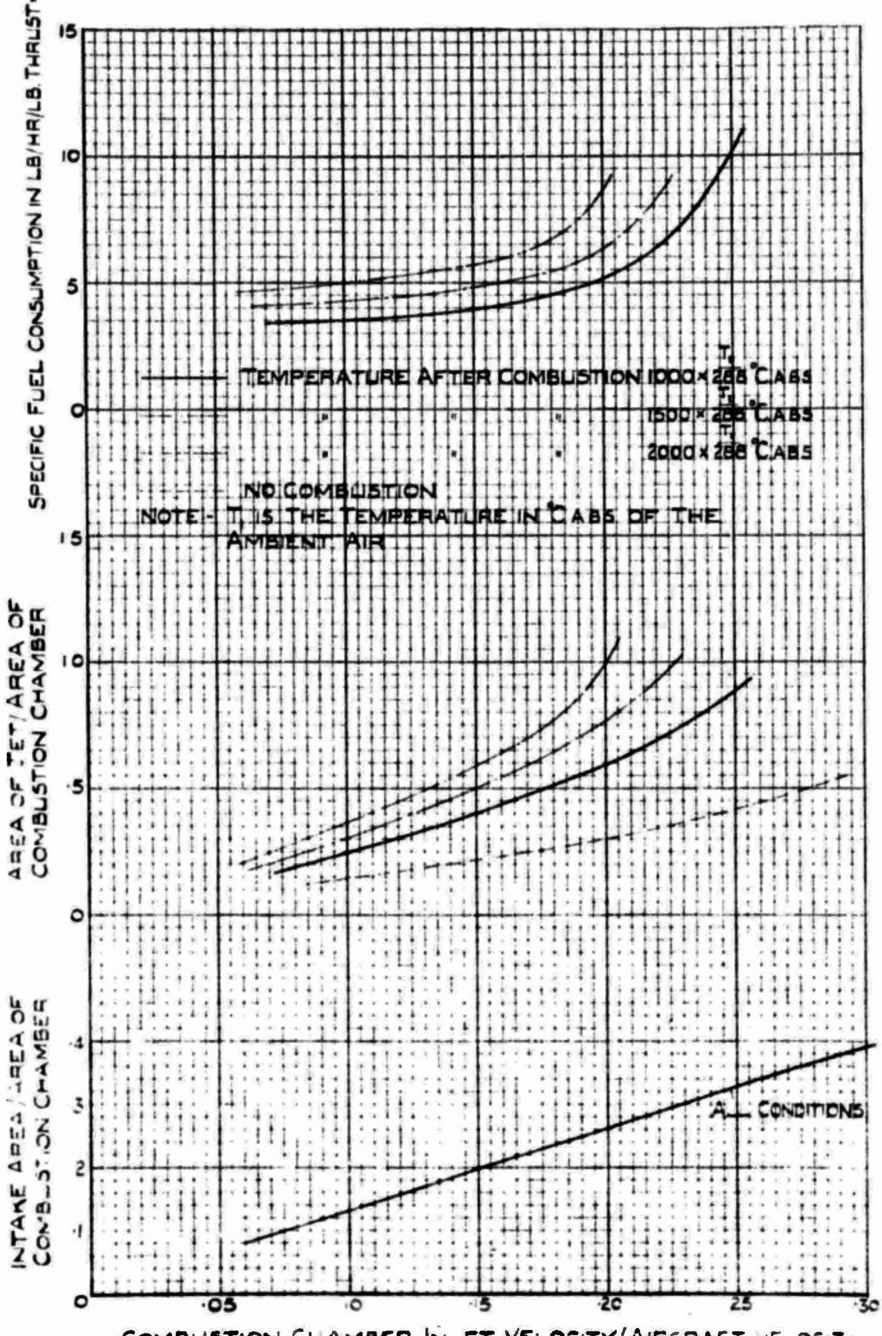
British MOST SECRET and SECRET equals United States SECRET

AIRCRAFT VELOCITY = $600 \times \sqrt{\frac{T_1}{288}}$ M.P.H.



20

PROPULSIVE DUCTS

British MOST SECRET and SECRET equals United States SECRET

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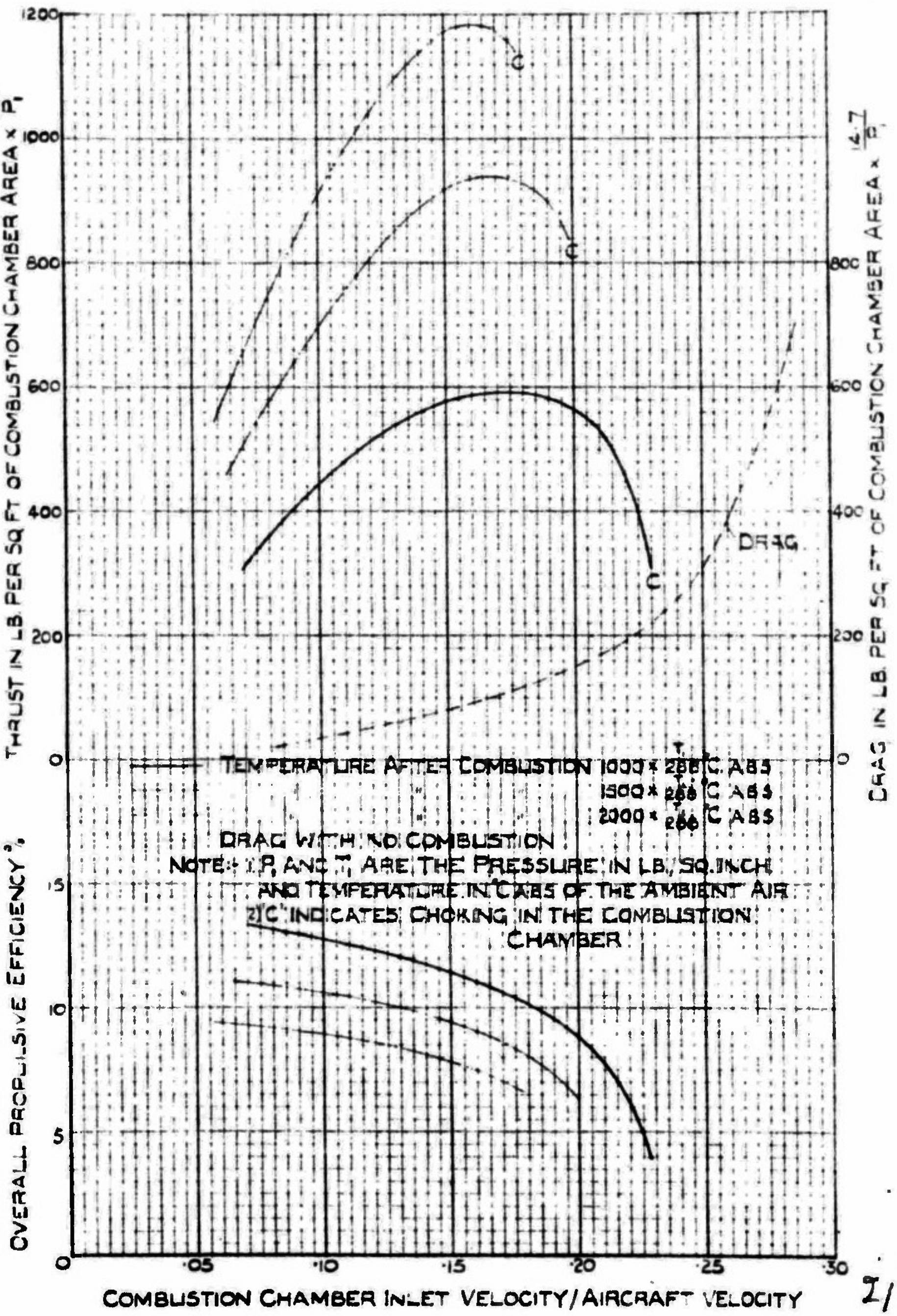
FIG. 10

AIRCRAFT VELOCITY = $900 \times \sqrt{\frac{T_1}{288}}$ M.P.H



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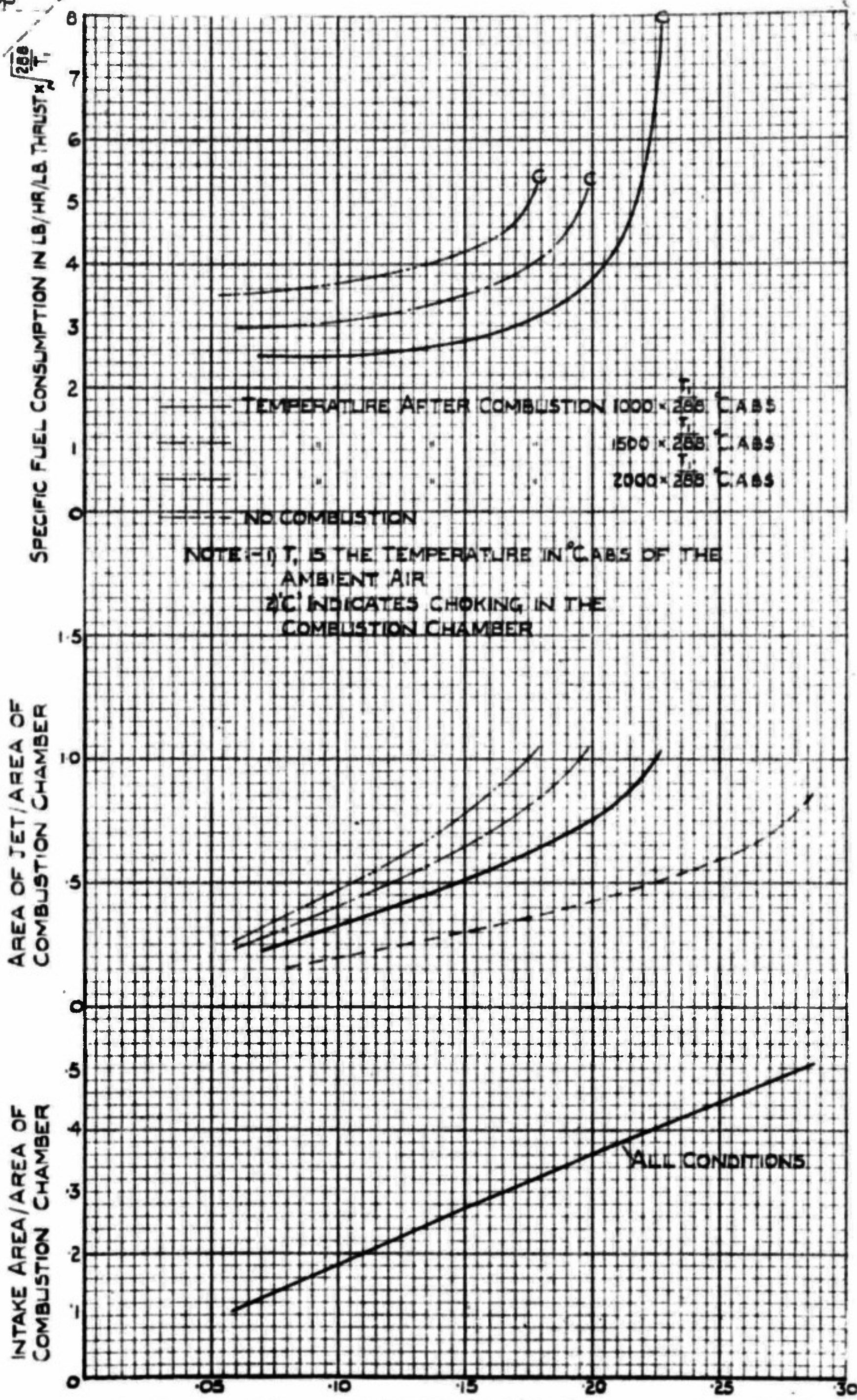
WA 2261 7

FIG. 11

AIRCRAFT VELOCITY = $900 \times \sqrt{\frac{T_1}{288}}$ M.P.H.



COMBUSTION CHAMBER INLET VELOCITY/AIRCRAFT VELOCITY

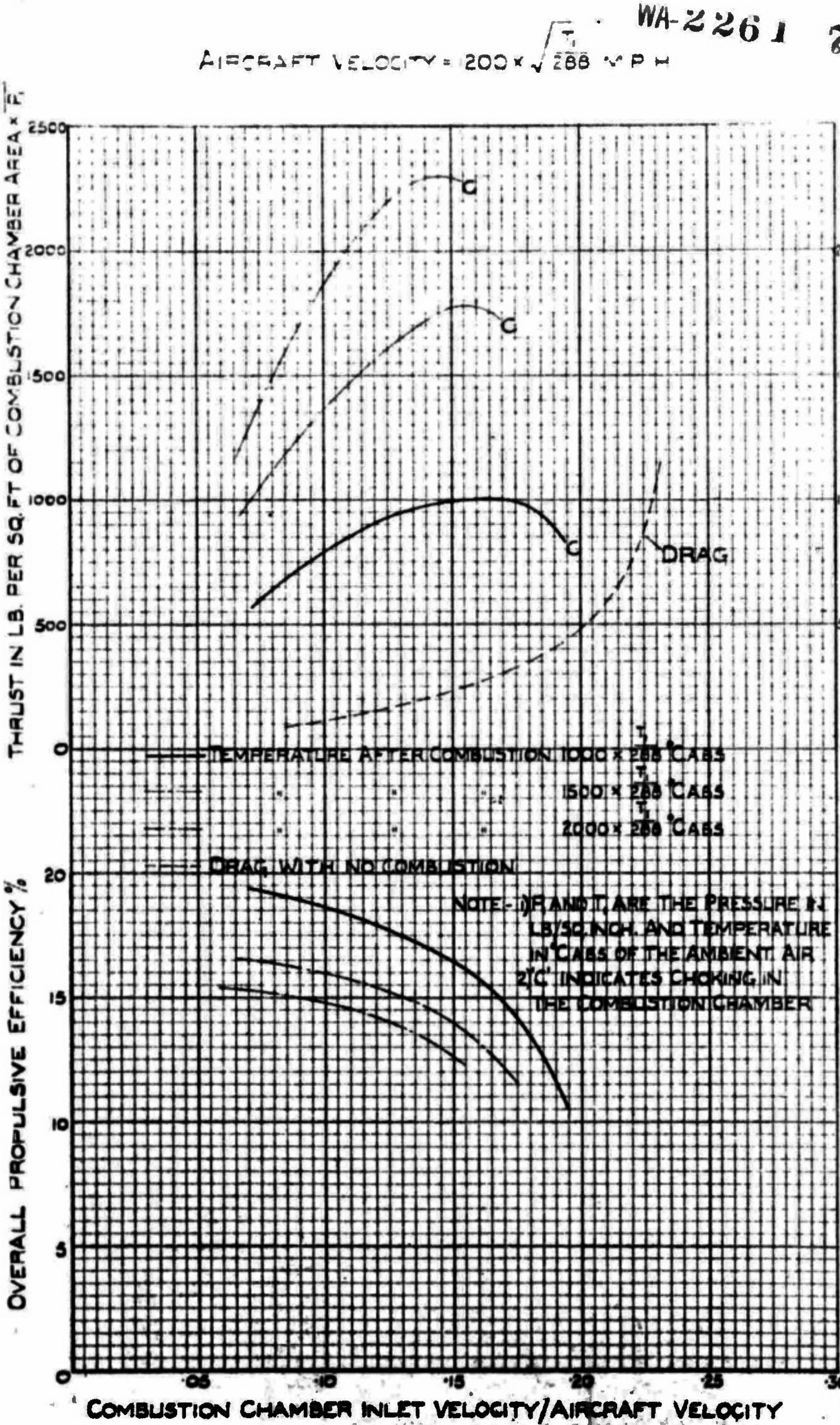
22d

PROPULSIVE DUCTS

23/5/43

FIG. 12

WA-2261 7



23 l

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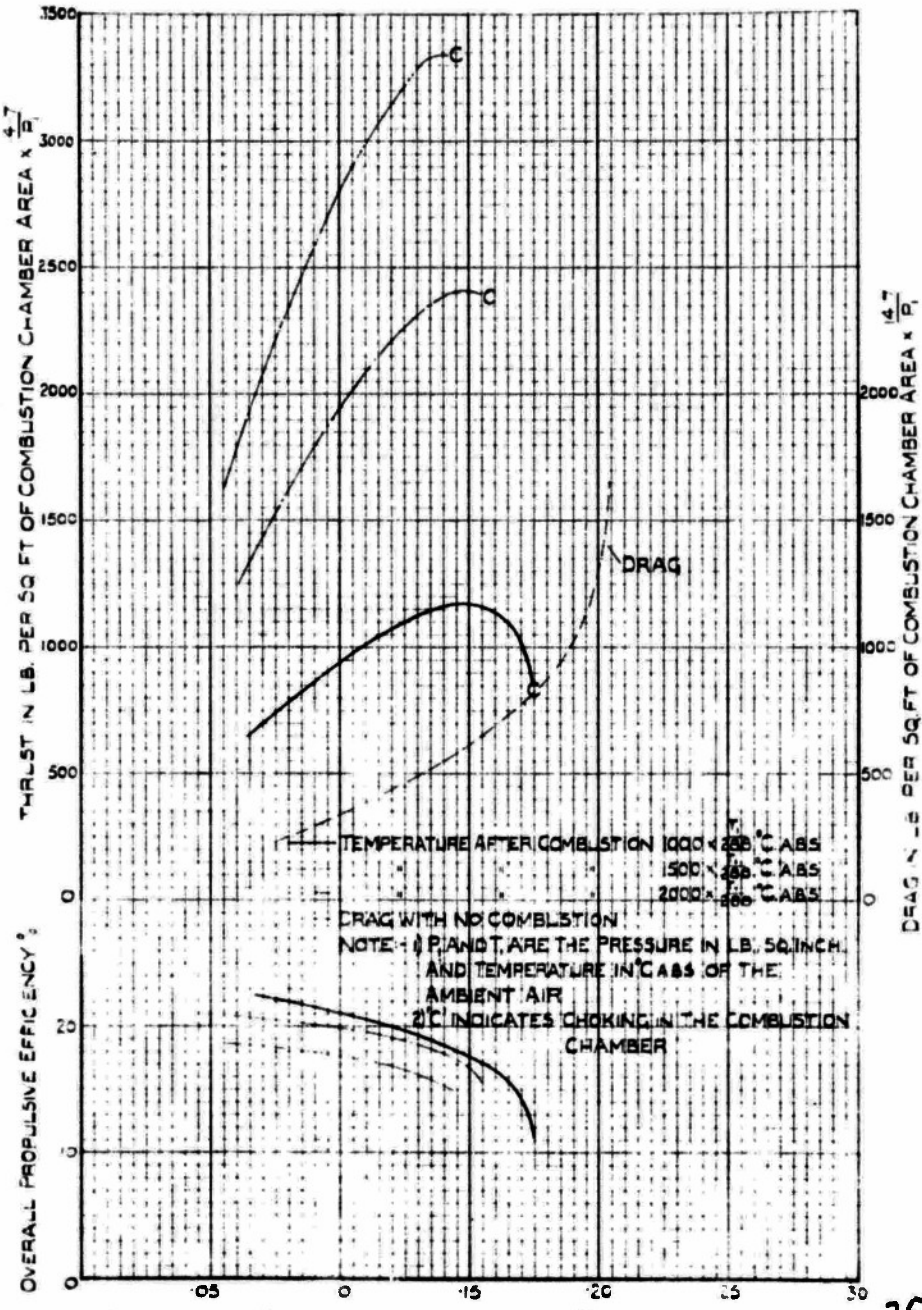
British MOST SECRET and SECRET
equals United States SECRET
British MOST SECRET and SECRET
equals United States SECRET

FIG 14

WA-2261 7

AIRCRAFT VELOCITY = $1500 \times \sqrt{\frac{T_1}{288}}$ M.P.H



PROPULSIVE DUCTS

25d

British MOST SECRET and SECRET
equals United States SECRET

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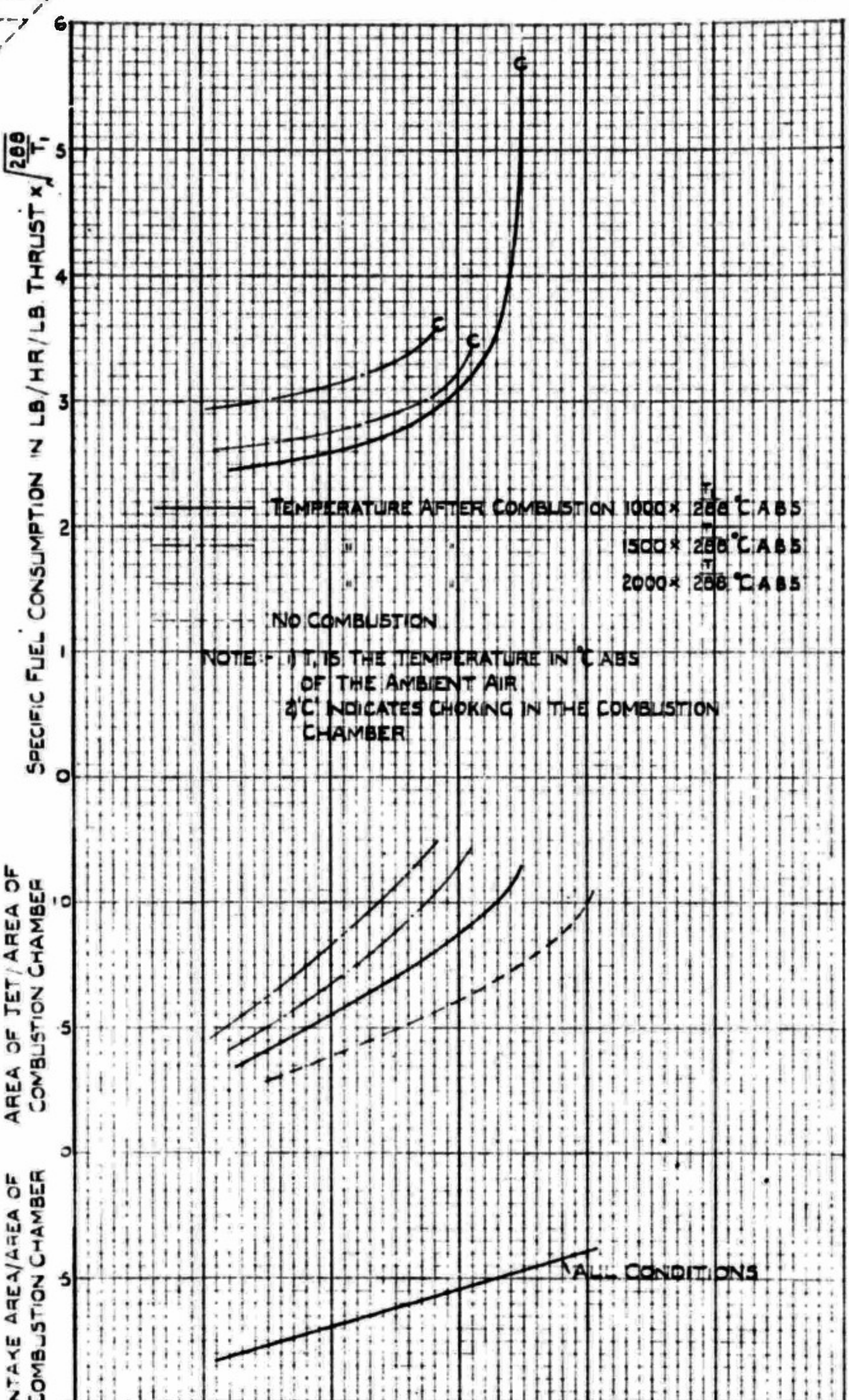
FIG 15

AIRCRAFT VELOCITY = $1500 \times \sqrt{\frac{T_1}{288}}$ M.P.H.



PROPULSIVE DUCTS

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TITLE: Note on the Performance of Propulsive Ducts

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ABSTRACT:

In order to estimate the over-all performance of a ramjet engine, various calculations must be conducted on the aero and thermodynamic factors of the air intake, combustion chamber, and exhaust nozzle. These calculations are shown and explained, and the results are given. A schematic drawing of a ramjet engine and the nomenclature of the symbols are included. Previously determined calculations are discussed.

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DIVISION: ~~Power Plants, Jet and Turbine (5)~~ 37

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